RAVEN-2

Around-The-World UAV Project

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INTRODUCTION

The Raven around-the-world UAV project is part of an on-going effort to build up a significant European capability in the design, construction and operation of large UAV's and manned reconnaissance aircraft.

The goal of the project is to fly a large high-altitude jet UAV non-stop and un-refuelled around the world using the trans-polar route.

The project will demonstrate the technology of long-range reconnaissance UAVs. It will develop the procedures and capability of operating large UAVs from conventional air bases in the conventional air traffic environment. It will establish an industrial grouping of companies capable of becoming prime-contractors for future military UAV procurements.

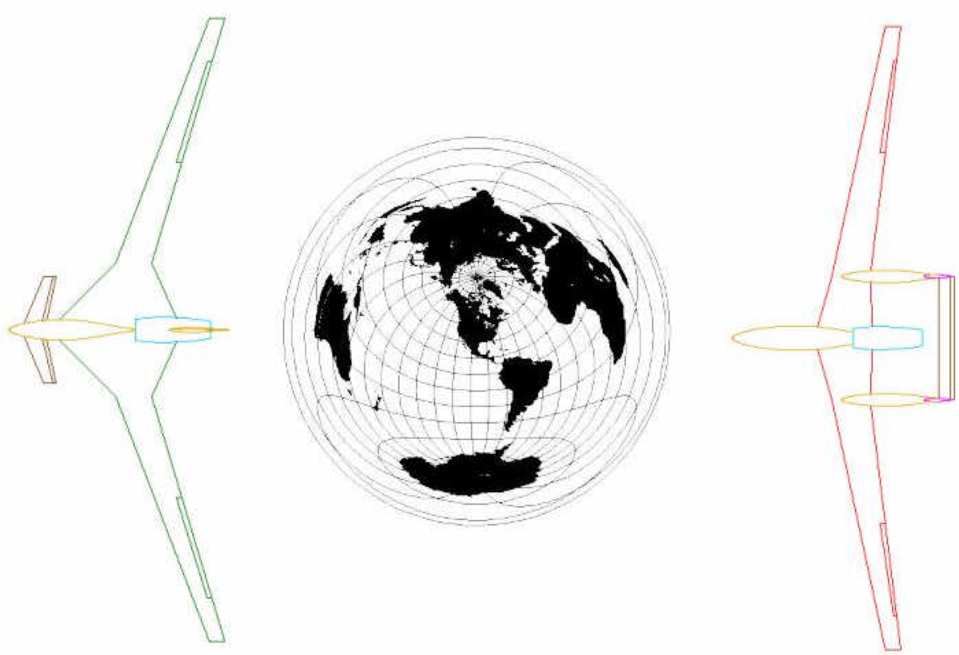

Figure 1. Raven-2 Project

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PERSONAL SUMMARY

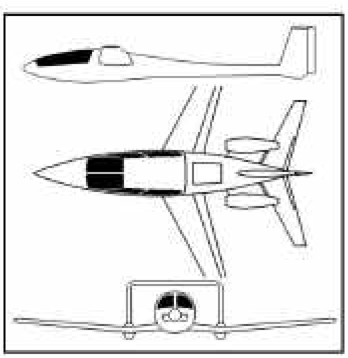
Chris Burleigh, M.Sc., C.Eng., MRAeS.
Aircraft Engineering Consultant,
Black Art Composites Limited, UK.

Chris Burleigh entered the aerospace industry in 1973 as an apprentice with Hawker Siddeley Aviation Ltd., Kingston-upon-Thames. He completed both the shop floor "craft" and the engineering "technician" training schemes before joining the airframe design office. His education during this period progressed through an O.N.C. to an H.N.C. at Kingston Polytechnic followed by a Master of Science in Aircraft Design at the Cranfield Institute of Technology.

Chris's work at Hawker Siddeley (later BAe) included structural layout and detail design of components for the YAV-8B and the AV-8B Harrier II aircraft. He later joined the composite materials development group at Kingston where he participated in the design and testing of carbon fibre airframe components. From 1982 to 1984 Chris was employed at Canadair Ltee., in Montreal where he first became involved in UAV projects.

After returning to the UK, Chris worked for a period as a contract stress engineer at BAe Warton on the E.A.P. experimental aircraft and at BAe Kingston on the single-seat Hawk-200. He then joined Edgley Aircraft to participate in Type Certification of the Optica light aircraft and became a CAA approved stress and design signatory. He went on to become Chief Designer for the Brooklands Aerospace Group were he continued with development of the Optica, including Australian and US Type Certification programs. Chris joined Chichester-Miles Consultants in 1990 where he became Chief Designer for the CMC Leopard jet project. In 1998 Chris joined Burt Rutan's Scaled Composites Inc., in Mojave, California as an aircraft structures engineer. He participated in a number of exciting projects including the Roton space rocket, the Proteus high-altitude aircraft and a NASA high-altitude UAV.

In 2001 Chris returned to the UK to establish his own composites and aerospace consultancy and engineering company, Black Art Composites Ltd. He is currently providing design engineering services to UK aircraft companies and developing UAV and aircraft technologies.



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RAVEN-2

Part 1. Around-The-World UAV Project

THE OBJECTIVE

The objective is to fly a jet powered UAV non-stop around the world without refuelling.

THE PURPOSE

1. To develop the capabilities and technologies of the participating companies and organisations. The long-term aim is to build up a significant European capability in the design, construction and operation of large UAV's and manned reconnaissance aircraft.
2. To demonstrate the technology of long-range reconnaissance UAVs.
3. To develop the procedures and capability of operating large UAVs from conventional air bases in the conventional air traffic environment.
4. To establish an industrial grouping of companies capable of becoming prime-contractor for future military UAV procurements.

PROJECT REQUIREMENTS

1. Flight performance must be adequate to demonstrate the potential usefulness of long range UAV technologies in the military reconnaissance environment.
2. The design of the UAV shall be such that read-across to a potential military application is maximized.
3. The flight shall begin and end at a conventional airfield in the UK or mainland Europe.
4. The UAV shall operate autonomously but have the capability for redirection at any point during the mission. Take off, recovery and landing may be assisted by remote piloting.
5. Communication of performance, navigation and research data to and from the UAV shall be possible throughout most of the flight.
6. Navigation and collision avoidance systems and procedures shall minimize the risk to other airspace.
7. The overflying of populated areas and the risk of air traffic conflicts are to be minimized.

THE CHALLENGE

The only aircraft that has ever flown around the world without refuelling is the Rutan "Voyager", flown by Dick Rutan and Jeanna Yeager. This feat was achieved in 1986. No jet powered aircraft and no UAV has ever matched it. The US Air Force "Global Hawk" UAV has demonstrated ferry flights in excess of 14,000 km and manned military aircraft have exceeded that distance with air-to-air refuelling. They have not however come close to achieving the 41,000 km range needed for a flight around the world.

THE ROUTE TO BE FLOWN

The trans-polar route around the world selected for the project is shown in Figure 2. It has a number of advantages over flying an equatorial route. Most importantly, it avoids over-flying any populated areas, with the exception of Alaska. The number of national air traffic control regions to negotiate is minimized and permissions to conduct the flight will be easier to obtain. The detour required to join and leave the selected route from a base in the UK or Europe is minimized.

The flight will be conducted at high altitude, above the flight levels used by commercial airliners.

Figure 2. Trans-Polar Flight Around The World

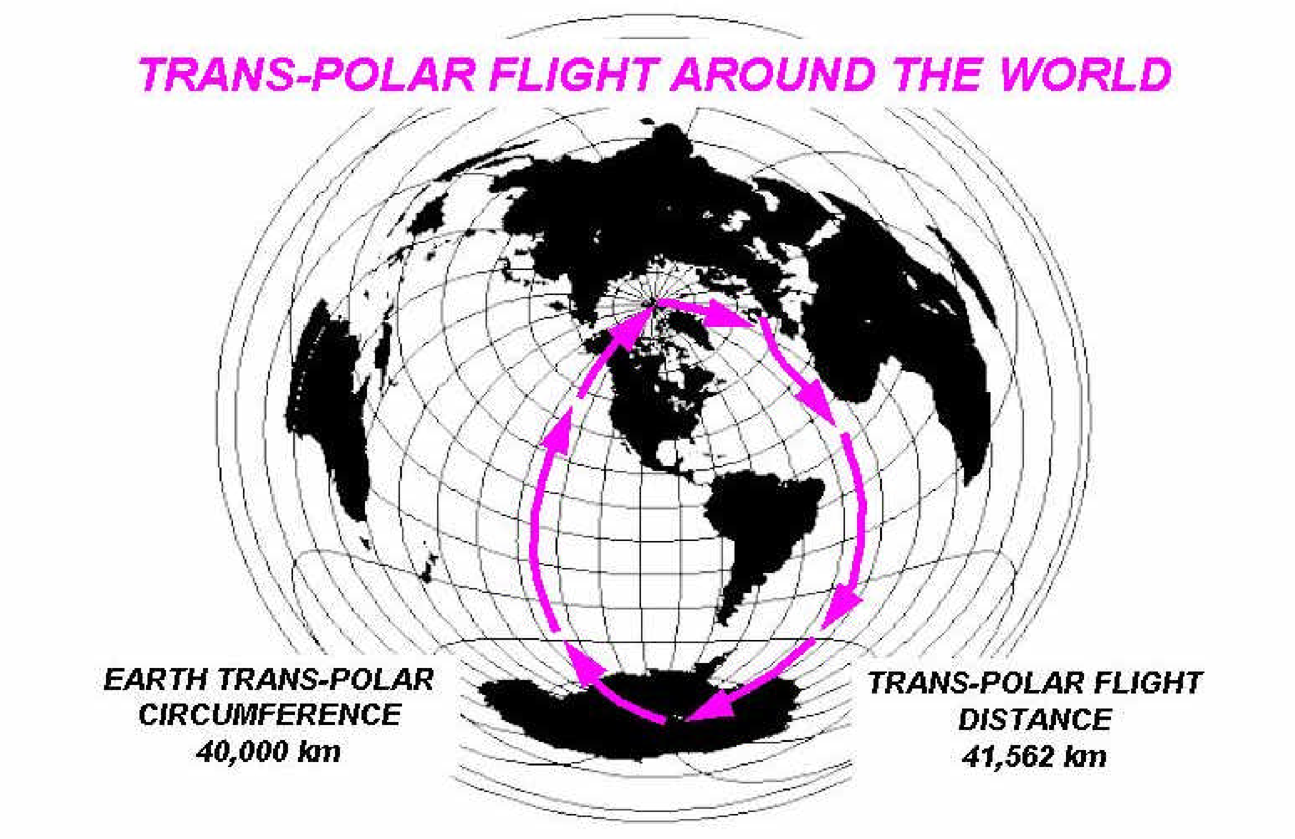


Table 1. TRANS-POLAR FLIGHT WAYPOINTS

Leg 1.	St Mawgan	to	West of Spain
	50.5^o N, 5^o W	to	43^o N, 10^o W
	Distance 1002 km.		
Leg 2.	West of Spain	to	Madeira
	43^o N, 10^o W	to	30^o N, 20^o W
	Distance 1822 km.		
Leg 3.	Madeira	to	South Georgia
	30^o N, 20^o W	to	55^o S, 45^o W
	Distance 9845 km.		
Leg 4.	South Georgia	to	South Pole
	55^o S, 45^o W	to	90^o S, 45^o W
	Distance 3889 km.		
Leg 5.	South Pole	to	Tahiti
	90^o S, 150^o W		to 17^o S, 150^o W
	Distance 8112 km.		
Leg 6.	Tahiti	to	Anchorage
	17^o S, 150^o W		to 62^o N, 150^o W
	Distance 8778 km.		
Leg 7.	Anchorage	to	North Pole
	62^o N, 150^o W		to 90^o N, 150^o W
	Distance 3111 km.		
Leg 8.	North Pole	to	West of Ireland
	90^o N, 11^o W	to	51^o N, 11^o W
	Distance 4334 km.		
Leg 9.	West of Ireland	to	St. Mawgan
	51^o N, 11^o W	to	50.5^o N, 5^o W
	Distance 669 km.		

The total flight distance 41,562 km. To this distance must be added reserves to allow for diversions, adverse weather and off-optimum flight conditions.

NOTE: The islands noted as waypoints above are used to describe the route and will not be directly over flown. Island based navigation aids can be used to confirm position during the flight.

AERODROME SUITABILITY

We are reviewing here, just UK airfields. In the broader European context a base on the west coast of Spain or Portugal would shorten the detour needed to join the trans-polar route and permit the climb-out and descent to be conducted further away from busy commercial airways. Table 2 summarizes the suitability of the airfields in the region.

Airfield Requirements:

1. Southwest UK location to minimize transition distance to join trans-polar meridian track.
2. Minimize over-flight and proximity to built-up areas.
3. Minimize obstructions and difficult terrain on departure and arrival.
4. Runway length must be more than 1000m, paved. Longer runways a big advantage.
5. Must have potential for accepting test and development flying.
6. Consider existing range safety, zone protection and air traffic situation.
7. Existing radar coverage and navigation aids.

Table 2. Airfield Suitability

AERODROME	RUNWAY LENGTH (m)	REMARKS
RAF St. Mawgan	2325	Best choice for location, runway size, surrounding airspace and environment.
Llanbedr MoD	2286	Has existing range safety for unmanned targets. Adds 185 km with Irish Sea return.
Swansea (civil)	1472	Adds 167 km.
Haverford West (civil)	1524	Minimal facilities. Adds 111 km.
RAF Chivenor	1833	Poor approach. Close to built up area. Adds 93 km.
RAF Mona	1666	Adds 222 km. Close to RAF Llanbedr unmanned target range.
RNAS Culdrose	1830	Close to built up area.
Caernarfon (civil)	1076	Short runway. Adds 222 km. Close to RAF Llanbedr unmanned target range.
RNAS Yeovilton	2310	Over-flight issues. Adds 370 km.
Boscombe Down MoD	3212	Over-flight issues. Adds 463 km.
RAF Valley	2290	Very busy.
RAF St. Athan	2745	Too close to Cardiff, inside CTR zone. Adds 296 km.

DEPARTURE AND RETURN

Following take off and during the climb to cruise altitude the UAV will pass through airspace used by commercial air traffic. A flight plan and departure route can be determined in advance, with the cooperation of the air traffic controllers that will avoid disruption of air traffic and minimize the risk of conflicts. On the day of departure the UAV can be held on the ground until the assigned route becomes available.

A business-jet will be used as a chase aircraft to escort the UAV through commercial airspace and give positive verification of its position and behaviour. The chase aircraft will have the capability to redirect the UAV and ensure that air traffic control directions are being correctly implemented. In the event of imminent danger to life or a major malfunction of the UAV the chase aircraft will have the capability to terminate the flight. It will stay with the UAV until it is safely out of commercial airspace and until the UAV's altitude and range exceed the capabilities of the chase aircraft.

On the UAV's return at the end of its mission, it may not be possible to positively confirm the quantity of fuel remaining and it may have suffered some undetected malfunction or damage. In addition, weather in the recovery area and at the landing site may have deteriorated. The recovery and approach shall be made over the sea so that if it unexpectedly runs out of fuel or the flight needs to be terminated it will not be a danger to people on the ground.

The UAV will report its position using a satellite data link. Ground based radar will detect its approach and transponder returns. The chase aircraft will be dispatched to make a rendezvous with the UAV as it descends towards the flight levels used by commercial traffic. Once contact is established the chase aircraft will escort the UAV back to the landing area. The chase aircraft will redirect the UAV around bad weather and serious icing conditions.

NAVIGATION

The main navigation systems that can be considered for autonomous operation of the UAV are GPS, ground based radio navigation aids and inertial reference. It was noted in a recent flight over the North Pole by Mike Melvill in the Proteus that GPS coverage was lost some distance short of the pole and the same is possibly true at the South Pole. A magnetic compass will be of no value over the polar regions and the poles are beyond the range of ground based navigation beacons. A solar compass or celestial navigation would work but may be difficult to implement on the UAV.

A navigation system based on inertial guidance would seem the most practical solution with confirmation of waypoints provided by GPS and radio navigation aids. Further confirmation of position could be obtained whenever the UAV passes within radar range of land.

COLLISION AVOIDANCE

The risk of collision is minimized by flying at altitudes well above commercial airline traffic and by selecting a route that avoids over-flying populated areas. Passive

collision avoidance is provided by the use of radar transponders and a TCAS system to alert air traffic controllers and other pilots to the presence of the UAV. The UAV will be clearly visible during the daytime and will display anti-collision strobe lights and position lights at night.

The UAV will be fitted with an air-to-air radar system to detect other aircraft. It will be programmed to steer normal aviation collision avoidance and separation procedures should it detect another aircraft at or near its flight level.

COMMUNICATION

The UAV will remain in satellite communication with its base crew throughout the entire flight. Commercial communications over the entire route appear to be possible using INMARSAT or IRIDIUM. Bandwidth on these systems is limited which prevents direct real-time operation and control of the UAV and its systems. Direct voice communication is possible and computer files can be downloaded to transfer data. The following tasks need to be addressed.

1. The passing of navigation information, changes of flight plan and systems control and management instructions to the UAV.
2. The passing of position information, weather information and aircraft condition data from the UAV to the base crew.
3. Relaying of air traffic control communications in real-time between the UAV and the base crew.
4. Transmission of research data measurements made during the flight.

FLIGHT MANAGEMENT

Preparation of the UAV for flight and sign-off for airworthiness will be the responsibility of the project engineering manager. Once he is satisfied that all is ready he will hand over responsibility to the flight director who will acknowledge satisfaction with the state of preparation of the UAV and formally accept responsibility for the flight.

The flight director will be a test pilot with suitable qualifications and experience to operate in the UAV’s air traffic environment and to make any critical safety decisions that may be needed. He will be assisted by the flight engineering team which will contain the flight, systems and communications specialists needed to monitor and operate the UAV.

Part 2. Feasibility Study: Achieving Around-The-World Flight Performance

OBJECTIVE OF THE STUDY

The objective of the feasibility study was simply to determine whether it is possible to construct a practical jet powered UAV capable of achieving the range required to complete a non-stop flight around the world.

The study would go on to investigate some of the configuration options and practical limitations to produce an outline design of a UAV for the mission.

THE STARTING POINT

One of the requirements of the project is to produce a UAV design with maximum read-across to a potential military long-range reconnaissance air vehicle. It would be helpful therefore to start by looking at some existing aircraft both manned and un-manned that are currently used in this role. Some aircraft are compared in Tables 3a and 3b.

Table 3a. Aircraft Comparison

TYPE	MAX T-O (lb)	MAX FUEL (lb)	SPEED (kt)	RANGE (n.m)	ENDURANCE (hr)
RQ-4A “GLOBAL HAWK”	25,600	14,500	343	13,500	36
LOCKHEED U-2	40,000	20,000	373	4,000	12
PROTEUS	15,800	5,900	280		14
CANBERRA PR-9	54,950	22,000	375	3,153	8.4

Table 3b. Aircraft Comparison (Continued)

TYPE	FUEL %GROSS	SPAN (ft)	WING AREA (ft^2)	W/b (lb/ft)	W/S (lb/ft)	A_W (ft^2)EST.
RQ-4A “GLOBAL HAWK”	57%	116.2	540	220	47	1890
LOCKHEED U-2	50%	103	1000	388	40	3500
PROTEUS	37%	77.58	479	204	33	2156
CANBERRA PR-9	40%	67.83	1045	810	53	4180

All figures in the tables above are taken from published sources and may not be accurate or may not correspond to the actual figures used in military deployments. This comparison is confined to jet aircraft with the only large UAV in this class being the Global Hawk.

In the tables “W/b” is span loading, i.e. gross weight / wing span. “W/S” is wing loading, i.e. gross weight / wing area. A_W is estimated airframe “wetted” surface area.

The design parameters to watch for are:

1. The weight of fuel as a percentage of gross (maximum take-off) weight which is 57% for the Global Hawk and might be assumed to get as high as 67% for our specialized around-the-world UAV.

2. The span-loading (weight/wing span) which dominates the induced drag equation and is critical for high altitude performance at sub-sonic speeds.
3. The wing-loading (weight/wing area), which determines the lift coefficient and profile drag coefficient at any given airspeed and altitude.

The Scaled Composites "Proteus" manned high altitude long endurance aircraft is shown in Figure 3. It is in roughly the size and weight category being considered for the Raven-2 project. The Proteus is designed for a long endurance loiter mission and is not optimized for flying long-range missions efficiently.

Figure 3. Scaled Composites Inc. Proteus Aircraft

Some Proteus features to note:
1. Weight of fuel and equipment (main undercarriage and tail booms) is distributed along the wingspan to minimize wing bending for minimum wing weight.
2. The trim tab on the left front wing control surface operates at very low and difficult Reynolds Numbers at high altitude.
3. Torsional stiffness of the inner wing, between the tail booms and fuselage is critical in avoiding aero-elastic flutter.
4. The engines, Williams FJ-44's have given trouble-free operation to altitudes of 65,000 ft "straight out of the box" and demonstrate the simplicity and advantage of using jet propulsion at these altitudes.

FACTORS THAT DETERMINE FLIGHT RANGE

The trans-polar flight distance to be flown is 41,562 km. To this a margin of just over 900 km is added to bring the range required for the baseline design to 42,500 statute miles. Later reassessment of the flight route and reserves may cause this target range to be revised upwards. A parametric study was conducted to find a series of aircraft configurations that would be capable of achieving this flight range.

The classical Breguet range equation for jet aircraft is;

$$\text{Range} = (\text{airspeed}/\text{sfc})\text{x}(\text{Lift}/\text{Drag})_R\text{x}\log_{10}(W_0/W_e)$$

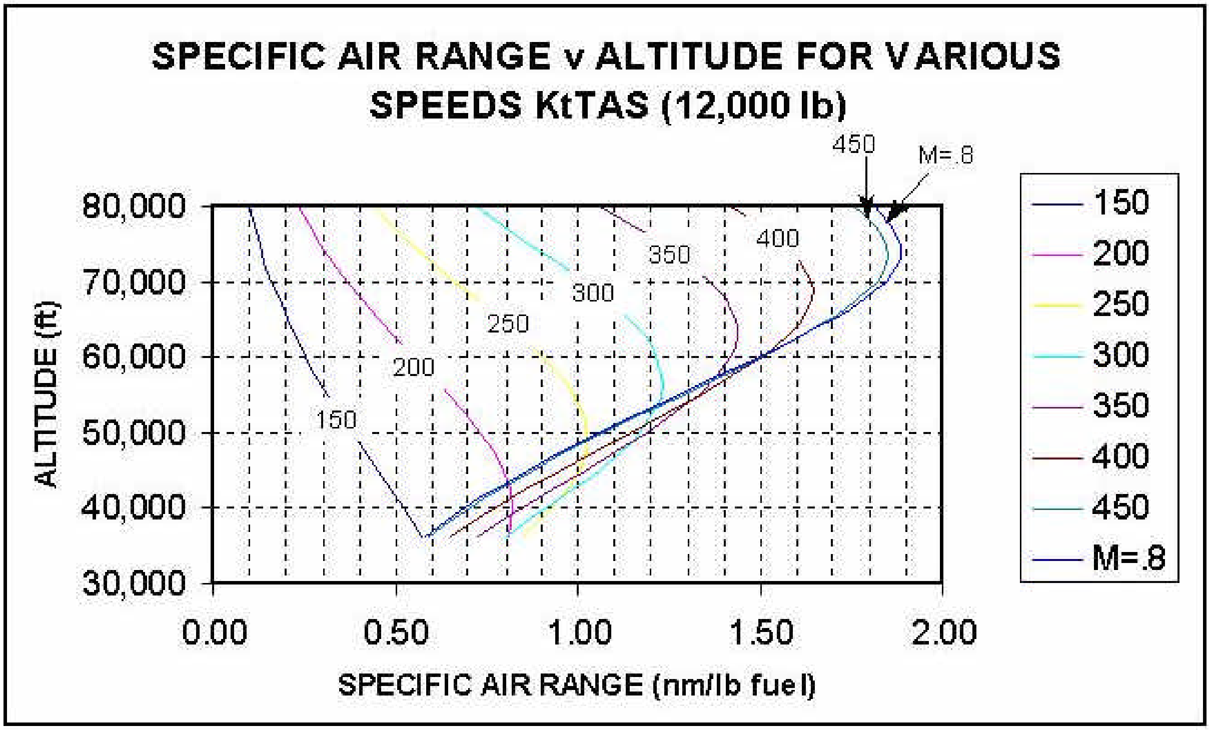
This all-encompassing equation is fine except that we do not know whether the optimum airspeed, specific fuel consumption and lift/drag ratio are constant throughout the flight. A method of calculating “specific air range” has been used in this study to assess the range performance of an aircraft at any point in its flight. This method allows for a more detailed assessment of the critical flight conditions as fuel weight, altitude and airspeed are varied throughout the flight.

$$\text{Specific Air Range} = (\text{True Air Speed}) / (\text{Drag x Specific Fuel Consumption})$$

This produces units of nautical miles per lb of fuel burnt i.e. nm/lb.fuel (or km/kg fuel).

Figure 4 is a typical plot of specific air range for a particular aircraft configuration and weight at various speeds and altitudes. It can clearly be seen that an optimum flight altitude exists for each speed and that higher speeds can result in greater range. The Specific Air Range achieved and the optimum flight points will change throughout the flight as the weight of the aircraft changes with fuel burn-off.

Figure 4. Specific Air Range v. Altitude



The configuration parameters required for calculation of the specific air range in the chart above are:

1. Aircraft weight at the particular point in the flight.
2. Wing span and span efficiency factor (for induced drag).
3. Wing area.
4. Airfoil lift and drag coefficients at appropriate Mach and Reynolds number.
5. Maximum cruise Mach number, as limited by airfoil characteristics and wing sweep angle.
6. Parasite drag wetted area and drag coefficient.
7. Specific fuel consumption.

For each aircraft configuration a series of specific air range charts was produced to cover its range of all-up weights from maximum take-off weight down to empty weight so that its performance as fuel weight reduces can be assessed. Once the aircraft's specific air range performance is known at each point in the flight the total flight range can be determined by integrating specific air range with respect to fuel weight.

It turned out that with the aircraft operating at its optimum altitude for any speed the Breguet range equation gave almost identical results to integration of the specific air range charts. However, the specific air range calculations were also telling us the optimum altitude to be flown, fuel flow rates and the best value of lift/drag ratio that could be achieved at any weight and speed.

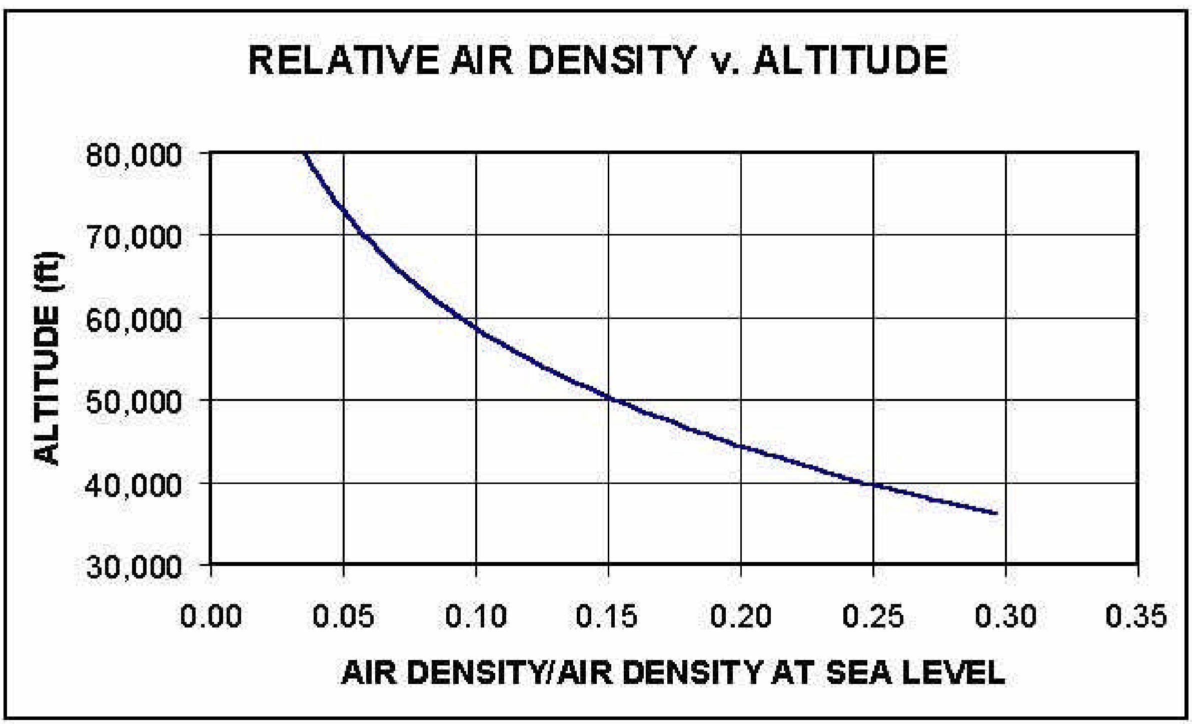
ALTITUDE LIMITATIONS

The best range performance for a configuration occurs at high altitude. As the aircraft flies higher at a constant true airspeed the air density is reducing, causing a reduction in dynamic pressure and indicated airspeed. This reduces the drag. At some altitude the aircraft will reach its best lift/drag ratio, if it continues to climb the drag will begin to increase again.

Increasing the wing area (reducing wing loading W/S) forces the aircraft to fly higher to achieve the best lift/drag ratio. As fuel is burnt off the aircraft gets lighter and its most efficient lift/drag ratio occurs at increasingly higher altitudes.

There is a practical limit to the maximum altitude that can be used because of the limitations of the engine. As altitude is increased the air density is reduced causing a reduction in the mass flow of air through the engine. The reduction of air density with altitude is illustrated in Figure 5. This reduces thrust, which in itself is not a problem provided that an engine with sufficient thrust at altitude has been selected for the aircraft but there may be problems keeping the engine running as air mass flow is reduced to a tiny fraction of its sea-level value. For this design study a maximum altitude limitation for the engine was set at 80,000 ft.

Figure 5. Air Density v. Altitude



AIRSPEED AND MACH NUMBER LIMITATION

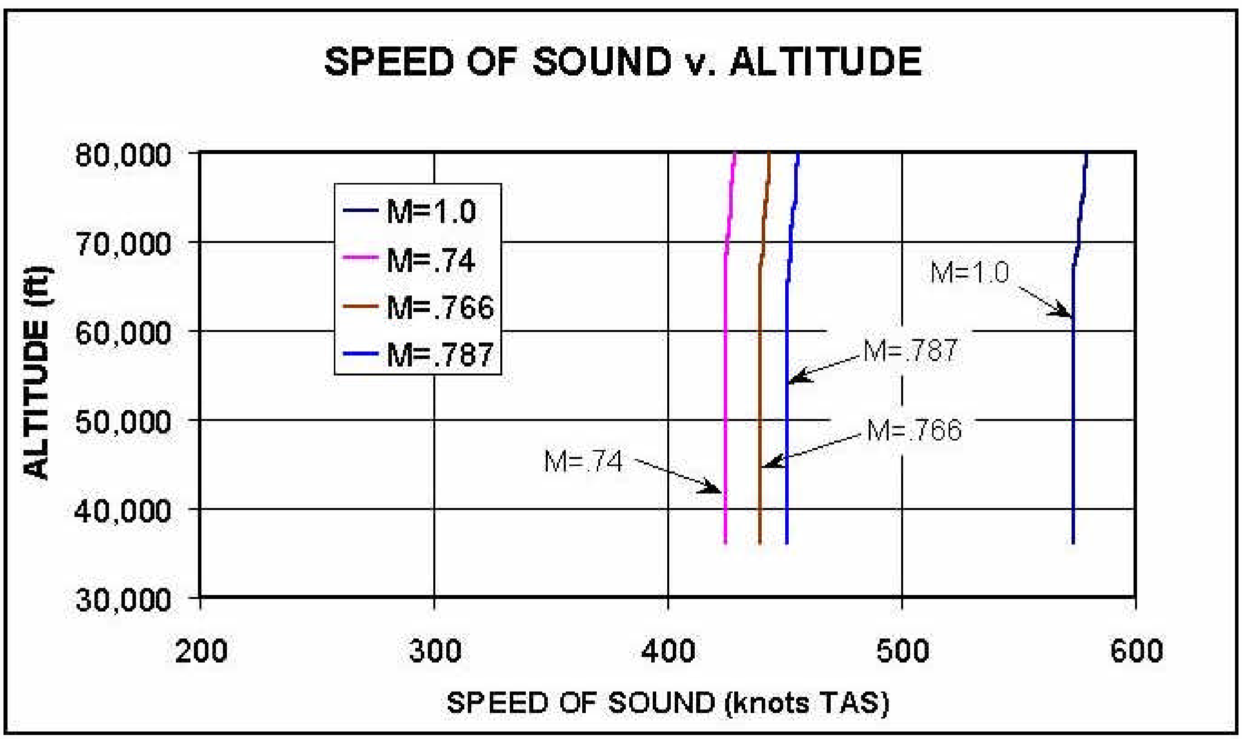
As a wing passes through the air the shape of the airfoil section accelerates the air over the wing surfaces causing a change in local air pressure to generate lift. Wing profile drag is caused by a combination of friction and air pressure forces pushing rearwards on the wing. A careful and detailed study of the airflow around airfoil shapes was conducted to determine the best shape to use for the particular requirements of this project.

The local velocity of the accelerated air passing over the airfoil shape must not reach the speed of sound. If this were to happen shock waves would form, greatly increasing the drag. The airfoil shape developed for this project will produce its best lift/drag ratio up to a flight Mach number of M=0.74 or 74% of the speed of sound.

Sweeping the wings backwards will allow the airfoil section to be used at slightly higher Mach numbers before shock waves begin to form. The M=.74 limited airfoil can be used at M=.766 with 15^{o} of sweep and at M=.787 with 20^{o} of sweep. At angles greater than 20^{o} drag of the airfoil may begin to increase as laminar flow over the surface becomes more difficult to maintain. Figure 6 shows the variation of speed with Mach number and altitude.

The higher airspeed permitted by the use of wing sweep is an advantage in improving some aspects of aerodynamic performance but has disadvantages in other areas. The wing structure will be more complex and heavier. The span-wise distribution of lift along the wing will not be optimum, resulting in an increase in induced drag.

Figure 6. Speed v. Altitude and Mach Number.



ENGINE PERFORMANCE AT ALTITUDE

The turbo-fan engine will give the best specific fuel consumption in the speed range we are considering. We need to look at engine thrust available at altitude. The optimum flight altitude may not be achievable if the engine cannot produce enough thrust. This gives us the option of flying at a lower, non-optimum altitude or installing a bigger and heavier engine with increased aircraft weight.

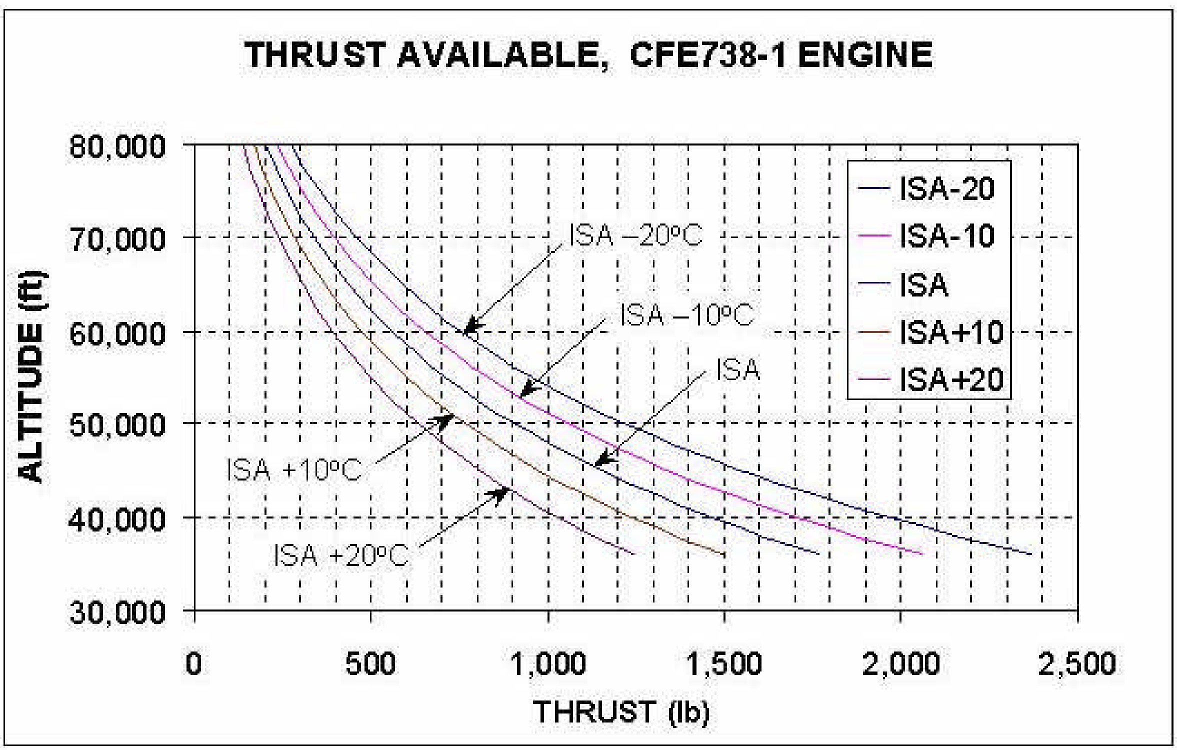
For the Raven-2 project we have selected the General Electric-Honeywell CFE 738-1 engine as the baseline powerplant. We do not have manufacturers high altitude performance data available yet and are therefore extrapolating its performance from the published data. The variation of thrust with altitude is shown below. Other engines that might be considered over a broad thrust range are the Williams FJ-44, Pratt and Whitney Canada PW-300 and PW-500 series and the Rolls-Royce (Allison) AE-3007.

Outline data for the engine:

Type: Two shaft turbo-fan
Bypass ratio: 5.3
Control system: Dual full authority digital engine controls (FADEC)
Dimensions: Length 99", width 43", height 48"
Dry weight: 1,325 lb (601 kg)
Sea-level static thrust: 5,918 lb (to 30^{o}C)
Cruise performance: 1,464 lb thrust and sfc=.64 lb/hr/lb.f at M=.8, 40,000ft.

Figure 7 shows the performance extrapolations for high altitude flight.

Figure 7. Engine Thrust v. Altitude



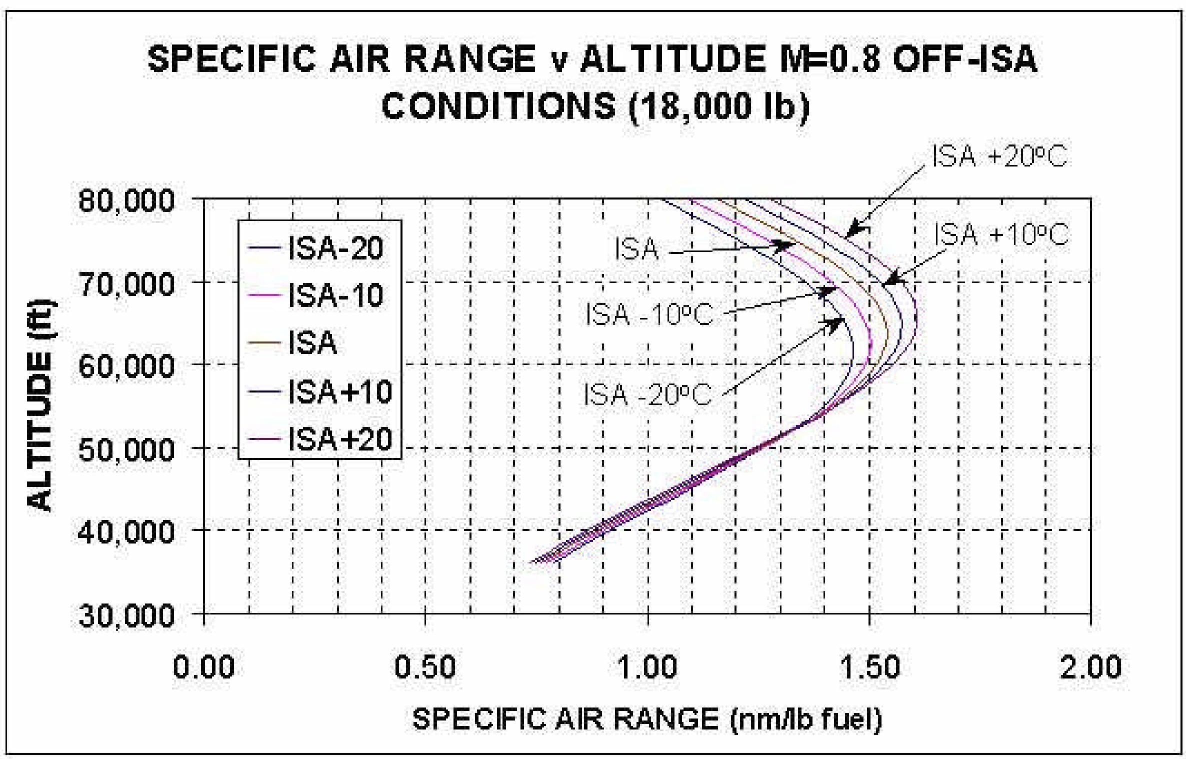
NON-STANDARD ATMOSPHERIC CONDITIONS

The performance work has been based on flight in the "International Standard Atmosphere" or ISA conditions and still-air. In the real world there are winds and atmospheric conditions that may vary considerably from ISA. The amount of variation can only be determined by monitoring daily weather reports over a long period of time. A statistical assessment can then determine the performance and fuel margins that will be required. Wind patterns will influence final selection of the flight route.

Weather conditions will be monitored during the flight both from aviation weather reports and from measurements made onboard the UAV. Adjustments to flight altitude and routing can be made to achieve the best range performance and make best use of the wind. The temperature variations affect air density, the speed of sound and engine performance. Figure 8 shows the combined affect of these changes on specific air range. It shows that it is an advantage to fly in warmer air.

Air temperatures in the polar and equatorial regions vary considerably from the ISA normal. Flying the trans-polar route around the world helps to average out these variations. Flying the equatorial route would keep the aircraft in the tropics where lower temperatures at high altitude are common. This would reduce the UAV's range performance.

Figure 8. Specific Air Range v. Altitude, Off-ISA Conditions.



UAV AIRCRAFT CONFIGURATION

A range of different configurations was systematically examined to determine the optimum configuration for the UAV. The achievable range was found by creating Microsoft Excel spreadsheet tables of specific air range at various weights for each UAV configuration. Cut-off limitations were applied for maximum Mach number, lift coefficient and engine thrust available. These values were integrated over the duration of the flight from maximum weight to empty weight to determine the total flight range.

An example of the parameters for a typical configuration are listed below:

Maximum weight = 18,000 lb Minimum weight = 6,000 lb
Wing area = 520 ft^2 Wing span = 103 ft
Parasite drag wetted area (wetted area excluding wing) = 500 ft^2
Coefficient of skin friction over parasite area = .005
Wing profile drag coefficient (at C_L=0.5) = 0.0067
Induced drag efficiency factor = 1.10
Engine specific fuel consumption = .64 lb/hr/lb.f
Maximum Mach number = 0.74 (straight wing)

The "parasite drag wetted area" multiplied by the "skin friction coefficient" represents the drag of all parts of the aircraft with the exclusion of the wing. This covers the engine nacelle, fuselage, tail surfaces and interference drag.

The Excel spreadsheet calculates the specific air range, drag, lift coefficient and margin of engine power available at each altitude for a range of speeds up to the limiting Mach number. The spreadsheet table is scanned to locate the highest possible value of specific air range within the limitations. The calculation is repeated for the same configuration but with fuel weight reduced in steps of 2,000 lb down to the minimum weight of 6,000 lb.

For each aircraft configuration examined, the best specific air range values at each weight are then tabulated as shown in Table 4 to calculate the total range achievable. Keeping all other parameters fixed, the wing span (aspect ratio) is varied until the desired range is achieved. This process is repeated for different sets of parameters until a series of successful configurations is generated.

Table 4. Range Spreadsheet Summary

WEIGHT (lb)	SAR (nm/lb)	RANGE (nm)	ALTITUDE (ft)	DRAG (lb)	C_L	L/D RATIO	TAS (kt)	sfc lb/lb/hr	RANGE EQUN.	TIME (hours)
18000	1.144	0	56,000	571	0.5	31.52	424	.64	0	0
16000	1.288	2,432	58,500	507	0.5	31.56	424	.64	2,458	5.8
14000	1.464	5,184	61,000	446	0.5	31.39	424	.64	5,240	6.56
12000	1.719	8,367	64,500	380	0.5	31.58	424	.64	8,451	7.57
10000	2.052	12,138	68,000	319	0.5	31.35	425	.64	12,252	8.95
8000	2.593	16,783	73,000	253	0.5	31.62	427	.64	16,923	10.97
6000	3.466	22,842	79,000	190	0.5	31.58	428	.64	22,989	14.19
DISTANCE (km) =		**42,303**							**TOTAL TIME (hr) =**	**54.04**

Some useful illustrative plots from this table are shown in Figures 9, 10 and 11. These are for one specific aircraft configuration. Similar plots were made for each successful configuration.

Figure 9. Specific Air Range v. Aircraft Weight

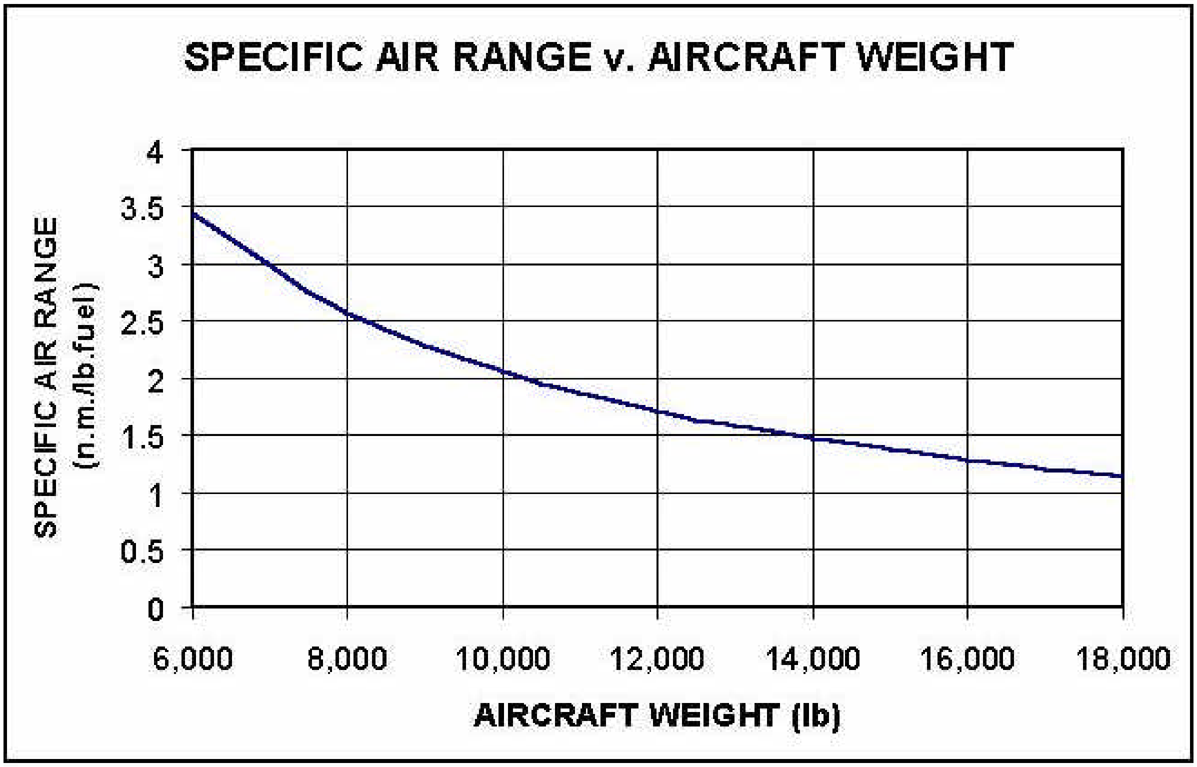


Figure 10. Drag v. Aircraft Weight

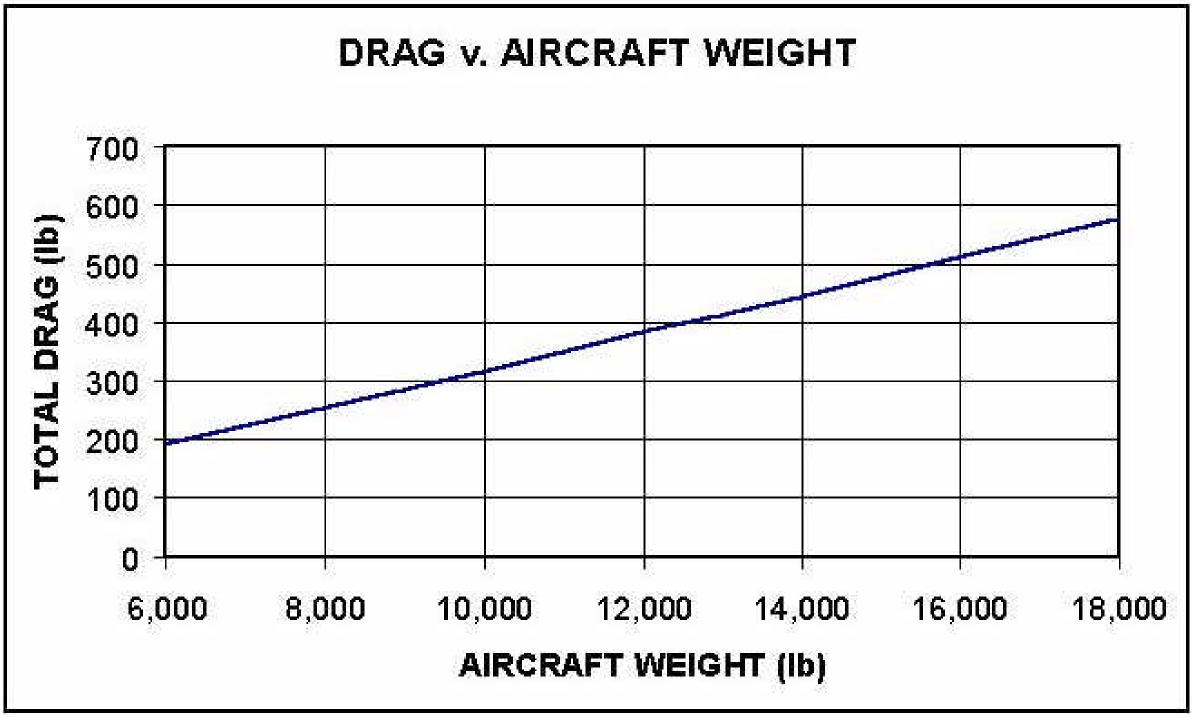
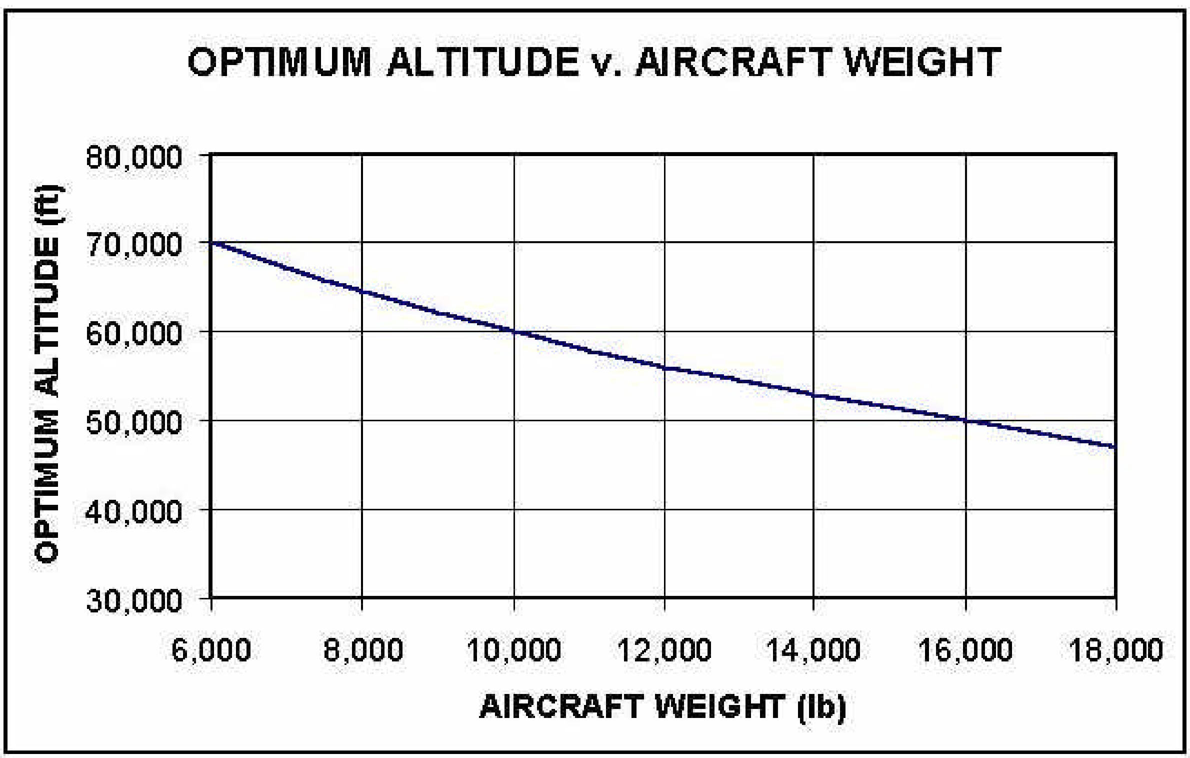


Figure 11. Optimum Altitude v. Aircraft Weight



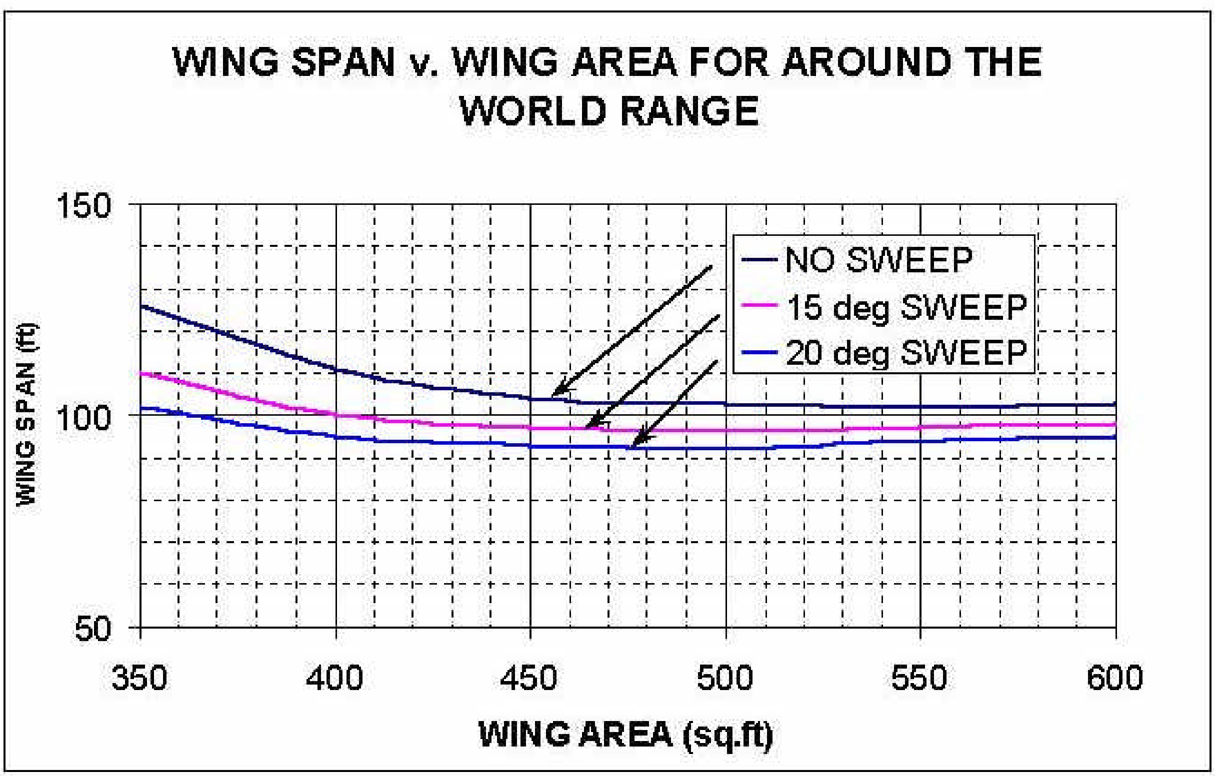
A summary of the successful configurations that will achieve the required range was collected from the spreadsheet results and is shown in Table 5.

Table 5. Geometry Optimization Summary

WING AREA (ft^2)	STRAIGHT WING			15° SWEEP			20° SWEEP		
	SPAN (ft)	POWER MARGIN	ASPECT RATIO	SPAN (ft)	POWER MARGIN	ASPECT RATIO	SPAN (ft)	POWER MARGIN	ASPECT RATIO
600	103	24%	17.68	98	7%	16.01	95	0%	15.04
550	102	30%	18.92	97	15%	17.11	94	0%	16.07
500	103	37%	21.22	96	21%	18.43	92	5%	16.93
450	104	43%	24.04	97	30%	20.91	93	15%	19.22
400	111	51%	30.80	100	37%	25.00	95	24%	22.56
350	126	57%	45.36	110	46%	34.57	102	35%	29.73

A range of wing configuration options is available from the results with the relationship between wing area and required wing-span being shown in Figure 12. The initial design parameters remain to be confirmed during the UAV preliminary design work. Refinement and adjustment of the values will be required as the design progresses.

Figure 12. Wing Span v. Wing Area



Part 3. Preliminary UAV Design Sizing

UAV AIRCRAFT INITIAL SIZING

The preliminary sizing investigation resulted in a range of wing configurations available for a non-stop flight around the world. All sizing calculations were based on one particular engine choice. If a different engine were to be used then the parametric study would be re-worked to generate a different series of possible configurations.

The basic UAV design parameters are:

Maximum weight = 18,000 lb (8165 kg)
Empty weight = 6,000 lb (2722 kg)
Parasite drag wetted area = 500 ft^2 (46.45 m^2)
Parasite drag friction coefficient = 0.005
Design lift coefficient = 0.5
Airfoil design drag coefficient = 0.00677

From the range of wing configurations available from the investigation two have been selected for further analysis. These two represent the extremes of the available range and will be developed into two different designs, one of conventional configuration and one of un-conventional configuration. In this way the advantages and disadvantages of the various features can be assessed.

Conventional Configuration *Raven 0-2*

Weights as above.
Straight wing (zero sweep)
Wing area = 500 ft^2 (46.45 m^2)
Wing span = 103 ft (31.39 m)
Cruise Mach No. = 0.74

Un-Conventional Configuration *Raven 2-3*

Weights as above.
20^o wing sweep
Wing area = 400 ft^2 (37.16 m^2)
Wing span = 95 ft (28.96 m)
Cruise Mach No. = 0.787

The wings have been selected to give a reasonable margin of available power and to give the flexibility to improve performance if it is found to be falling short of the target later in the development process. Range performance improvements can be accommodated in either configuration by simply adding more span onto the wingtips to yield an increase in both wing span and wing area. The power and altitude margin is important here because adding wing area will force the aircraft to fly higher to achieve its optimum range performance.

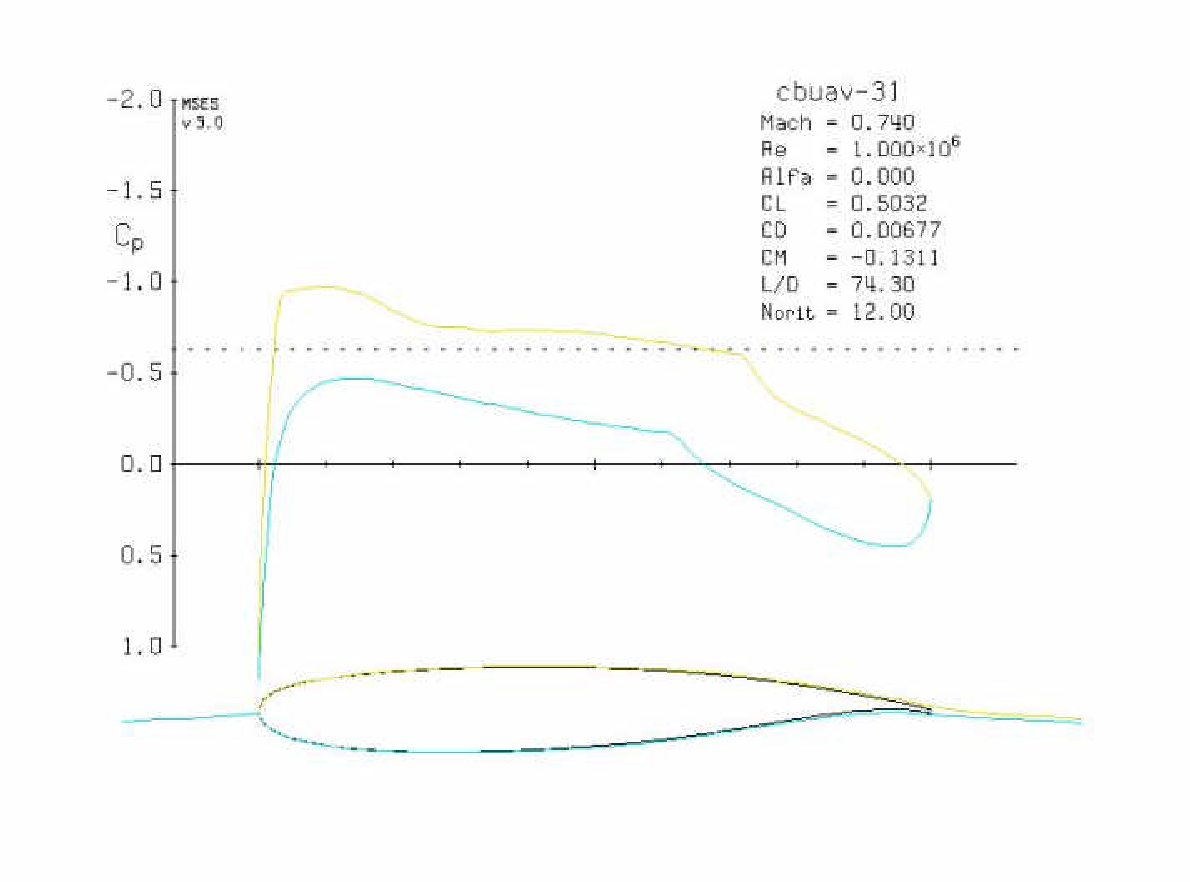
The UAV designs must achieve or better the parasite drag wetted area and drag coefficient of the basic sizing analysis. Once the UAV design has progressed and airframe drag can be properly estimated then the range analysis can be revised into a second iteration of the design.

WING AIRFOIL SELECTION

An independent airfoil analysis was conducted by Daniel Hatfield to determine the optimum airfoil for the project. The aim of his work was to achieve the best lift/drag ratio possible at the cruise Mach numbers required for the project.

The airfoil is shown in the Figure 13. Other features to note are that it is relatively thick at 12% and has a large internal volume. This will help to minimize structural weight and maximize the fuel volume.

Figure 13. Airfoil Section



maximize lift/drag ratio of the airfoil section. The tailless configuration offered no way of balancing out the pitching moment of the wing without significantly reducing the lift coefficient over a large part of the span. A separate stabilizing surface, either tail-plane or canard with a long moment arm is a more efficient way of balancing out the moments.

WING FUEL VOLUME

In order to get the most structural advantage from inertia relief it is desirable to put all of the fuel in the wing. The internal volume of the wing configurations was assessed and adjustments made to the plan-form to squeeze the fuel in.

It was assumed that the wing section would contain fuel from the leading edge back to 75% of the chord. This is the proposed rear spar location. Aft of this point the volume is used to house the ailerons, flying controls and other systems.

Wing taper was found to have a significant affect on fuel volume. Greater taper caused the root rib cross sectional area to increase resulting in greater internal volume. The taper ratio (tip chord/root chord) was set at 0.35 to give a good induced drag efficiency, good taper and not cause difficulties with high lift coefficient and low Reynolds number at the tip. A single large main wing would give greater fuel volume than a configuration like the tandem-wings of the Scaled Composites “Proteus”.

The volume required to accommodate the 12,000 lb of fuel needed for the mission is 240 ft^3. For the straight winged configuration the 500 ft^2 wing does not have sufficient fuel volume. One option would be to use a larger wing but that would force the aircraft to fly higher and the increased surface area of the wing would add weight. An alternative solution, adopted here is to increase the chord of the wing locally at the root. It is assumed that the same airfoil thickness to chord ratio is used and that lift coefficient is reduced by reducing camber at the trailing edge. This adds some wetted area to the airframe but the enlarged sections near the wing root add stiffness and structural efficiency in the most highly loaded region of the wing.

The 20^o swept wing has a basic area of only 400 ft^2 and will also clearly be significantly short of internal fuel volume. For this configuration it was decided to move towards a more blended wing and body combination. This was done by greatly increasing the root chord to 20ft at the centreline but leaving the central part, within the fuselage clear of fuel so that it could be used to accommodate equipment. This approach increases the wetted area of the wing plan-form but allows the parasite wetted area of the fuselage to be reduced and reduces the interference at the wing to fuselage joint.

FUEL CENTRE OF GRAVITY LOCATION

The horizontal tail surfaces need to be kept as small as possible to minimize their wetted area and drag. To do this, the tail load required to balance the wing pitching moment and the weight moment must be kept to a minimum. An important part of this is to ensure that there is not a massive shift of centre of gravity as fuel is burnt off.

To minimize centre of gravity shift with fuel burn the centroid of the fuel volume is set to coincide with the aerodynamic centre of the wing plan-form. In this way the fuel C of G will be the same with tanks full and empty. Fuel management will then ensure that the C of G does not vary too much from this point with partially full tanks.

The exact location of the aerodynamic centre (and the longitudinal stability neutral point) will need to be determined once the influences of the fuselage and horizontal stabilizer are known. For initial sizing of the “straight wing” aircraft the fuel C of G will be set to coincide with the 25% chord (¼AMC) point of the cranked wing plan-form. In the final version, the fuselage will shift the aerodynamic centre forwards and we will probably set the C of G nearer to 30% of the AMC.

Introducing slight sweep angles into the plan-form was used to make the fuel C of G coincident with the 25% chord point, as shown in Figure 14.

For the “swept wing” aircraft a similar technique was used. In this case it was intended that a “canard” surface would be used for longitudinal stability and pitch control and because the fuselage would be smaller it would have less influence on the location of the aerodynamic centre. For initial sizing, the fuel C of G was set just ahead of the aerodynamic centre of the cranked wing plan-form. Sweep of the outer wing was maintained at 20^{o} but sweep of the sharply tapered inboard wing panel was greatly increased. The swept wing, with its cranked planform and blended wing and body is shown in Figure 15.

Figure 14. Conventional Configuration UAV

(Modified Straight Wing)

RAVEN 0-2

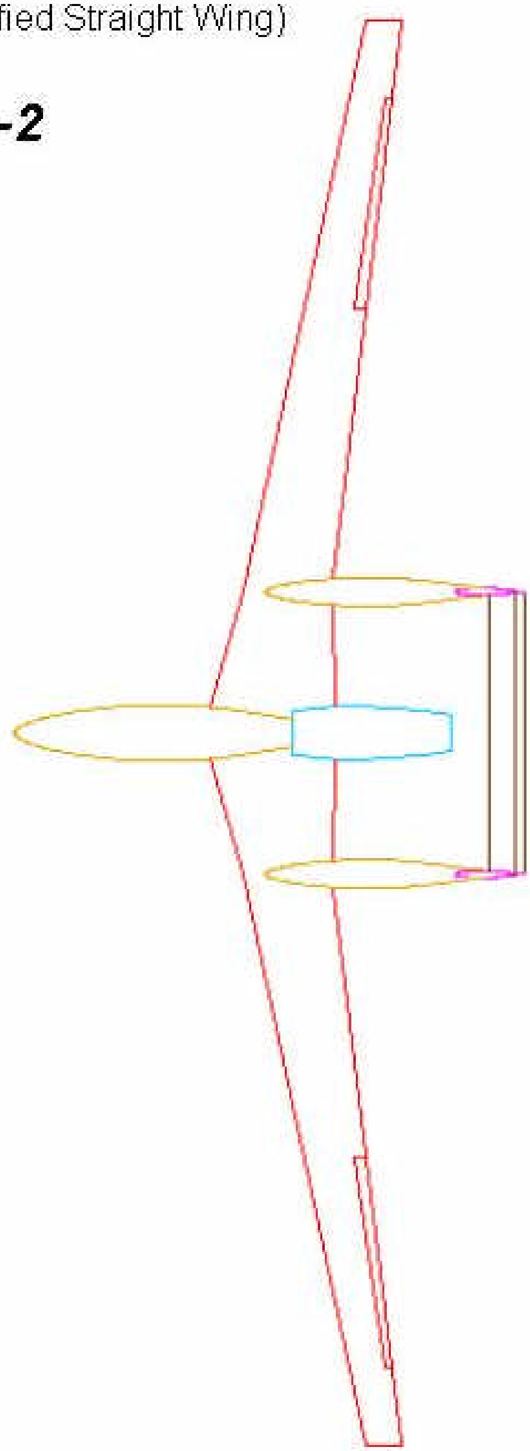
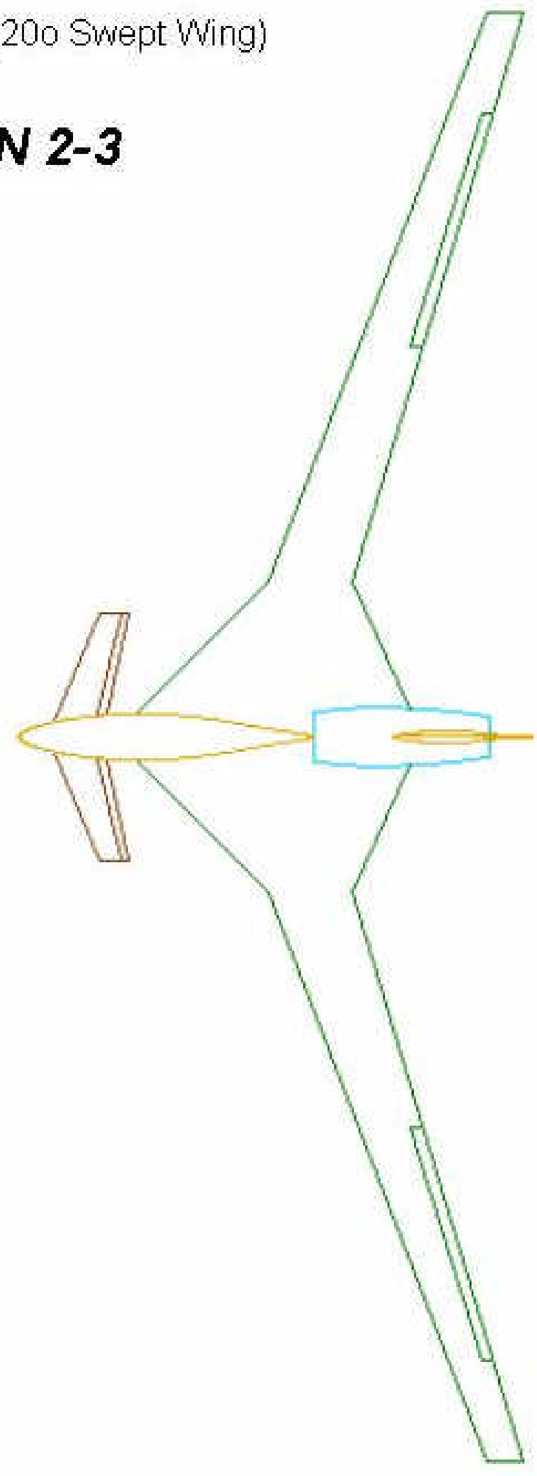


Figure 15. Canard Configuration UAV

(20o Swept Wing)

RAVEN 2-3



COMPARISON OF CONFIGURATIONS

1. Both configurations show the engine mounted in a pod with its intake above the rear part of the wing. This keeps the weight of the engine close to the C of G of the aircraft so that it can be balanced by the modest weight of equipment in the fuselage. The “straight wing” configuration has an excess margin of power of over 30% and it may be possible to use a smaller and lighter engine.
2. The “straight wing” is too thin to accommodate the main undercarriage. The undercarriage retracts into the tail booms. For the “swept wing” aircraft longer undercarriage legs are required to prevent the wing tips striking the ground. With the legs mounted at the wing kink rib the wheels will retract into the thicker wing roots.
3. The weight of the undercarriage and tail booms outboard on the wing of the “straight wing” configuration helps to reduce the bending loads in the wing.
4. The absence of a rear fuselage or tail booms on the canard configuration will save weight. The short tail fin moment arm will mean that the fin must be large and mounting it on frames around the rear of the engine nacelle will be fairly heavy. Overall though, the empennage arrangement on the canard aircraft will be significantly lighter than the conventional tail arrangement.
5. The overhung mass of the tail booms and empennage on the conventional configuration will require significant stiffening of the inner wing to avoid aero-elastic flutter.
6. The “swept wing” canard configuration is smaller and has fewer structural components. It should therefore be cheaper and lighter to build.

SOME SYSTEM DESIGN FEATURES

UNDERCARRIAGE

One of the recurring problems found in flight operations of the Proteus has been the occasional loss of oleo pressure or tyre pressure following extended periods of cold-soak at high altitude. To avoid damage to the UAV should this problem occur all undercarriage legs are to be fitted with dual wheel and tyre assemblies.

The brake assemblies will be rated with sufficient energy absorption capacity to stop the fully loaded aircraft in an aborted take-off at all speeds up to lift-off speed.

The undercarriage shall be designed for a maximum landing weight significantly below the maximum take-off weight to minimize the weight of the undercarriage units. Use of a single central main undercarriage leg plus balancing outrigger wheels may be considered as a way of reducing undercarriage weight.

The use of a separate take-off trolley, left behind when the UAV lifts off may be considered. In this way the undercarriage need only be sized for landing and braking at lighter weights. An auxiliary propulsion device might be fitted to the trolley to help accelerate the fully loaded UAV and shorten the take-off roll.

FLYING CONTROL POWER SYSTEM

It is envisioned that a dual circuit hydraulic powered flying control system will be used. The servo-valves of the flight control actuators will be operated by the electronic flight control system.

Hydraulic power will be provided by electric pumps with accumulators to maintain pressure during periods of high flow-rate. This will consume less power than continuously running engine driven hydraulic pumps. Batteries can maintain power to the pumps and flight controls in the event of the engine stopping.

ELECTRONIC EQUIPMENT CHAMBERS

Much electronic equipment is sensitive to the low temperature and low air pressure of high altitude flight. The UAV fuselage will consist of several sealed and pressurized chambers to house the electronics. The chambers will be purged with dry nitrogen or argon before take-off and on-board compressed gas cannisters will be used to maintain pressure. Cooling of the electronics will be by oil heat exchangers with the excess heat being transferred to the fuel in the wings.

ELECTRICAL POWER SYSTEM

The engine will be fitted with two alternators. Two separate electrical bus systems will be used with the potential to connect to the other alternator in the event of an alternator failure. An emergency bus will supply vital equipment to keep the UAV flying while electrical problems are being diagnosed, rectified or isolated. A 115V a.c. supply may be required for some equipment.

Battery power will be sufficient to keep the emergency bus and communications systems operating for at least as long as it takes the UAV to glide down to sea level. It must maintain sufficient reserve or have a dedicated supply to attempt to restart the engine.

END